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FINAL REPORT

ADVANCED STRUCTURAL REPAIR

Design and Installation of Bonded Repairs
to cracked fuselage skins on C-5A T/N 69-021

Submitted to the USAF European Office
of Aerospace Research and Development
and the
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ADVANCED STRUCTURAL REPAIR
FINAL REPORT

Design and Installation of Bonded Repairs
to cracked fuselage skins on C-5A T/N 69-021

Executive Summary

Two bonded advanced composite repairs were applied to cracked 7079-T6 aluminum fuselage skins on C-5A tail number 69-021 in October 1995. The repair method was based on extensive analytical work, fatigue experiments and proven in-field bonding techniques using advanced composite patches. This advanced bonded repair is considered a viable, low-cost alternative to replacing the cracked skin panels in the fuselage crowns of the C-5A fleet.

TABLE OF CONTENTS

1. INTRODUCTION

1.1. C-5A CRACKING PROBLEM

A typical sign of aging in the C-5A fleet is multiple small cracks in the upper (crown) section of the fuselage. The crown section experiences significant longitudinal tensile bending in addition to biaxial tension due to internal pressurization. Multiple short cracks (25 to 50 mm or 1 to 2 inches long) have been found in the crown of many aircraft. Most cracks are believed to have nucleated in the rivet holes of the 7079-T6 aluminum skins due to high fit-up stresses induced in manufacturing. Riveted repairs have been attempted without success; new skin cracks nucleate quickly from the corner rivet holes attaching the patch. Thus, the remaining repair alternatives are a bonded repair or reskinning the affected area.

The following pages discuss the development and installation of bonded composite repairs to cracks in the C-5A fuselage crown. Analytical studies involved three composite patch materials, including AS-4 carbon fiber/epoxy, boron/epoxy, and the S2 glass fiber/epoxy/aluminum laminate known as GLARE 2.

1.2. BACKGROUND OF CRACK PATCHING

"Crack patching" refers to the practice of applying adhesively bonded advanced composite repair patches to cracked metal structures. Crack patching was first successfully applied to the wing pivot fitting of the General Dynamics F-111 in 1970 [1]. The concept, first applied to cracked thick structures, has extended the useful lives of flawed airframes for many years when no other cost-effective alternative existed. In the past three years, severe cracks emanating from the "weep holes" in the lower wing skins of the C-141 fleet were repaired successfully with bonded boron/epoxy composite patches. Even more recently, the crack patching concept has been extended to the repair of cracked thin fuselage structures.

Many high-strength, stiff composite materials have quite low coefficients of thermal expansion (CTE) compared with aluminum alloys. Extensive analysis has shown that

moderate-CTE composite materials should perform better as fuselage skin repair materials than the higher stiffness, low-CTE composite patch materials. The moderate-CTE composite material GLARE has proven to be a viable alternative to the composite patch materials such as boron- and carbon-epoxy for pressurized fuselage skin repairs. (GLARE is a hybrid laminate of 2024-T3 aluminum and unidirectional S-glass/epoxy composite [2, 3]). The moderate-CTE patch materials give a lower stresses near the crack tip and reduce the adhesive shear strain. The lower crack tip stresses will result in extremely slow crack growth, while the reduced adhesive shear strain is important to ensuring a durable adhesive bond.

Previously reported tests showed crack patching with GLARE patches to be durable: that is, the patches remained intact for a reasonable number of aircraft fatigue lifetimes without allowing cracks to grow [4]. Disbonds or defects in the bond line between the patch and the skin do not dramatically change the behavior of the patched structure. The damage tolerance of the repair has been shown by documenting the beneficial effect on the structure regardless of damage that exists outside of the repair [5].

1.3. REPAIR ALTERNATIVES

Riveted repairs to crown skin cracking have already been attempted. Because of the high stress situation in the crown and the poor fatigue resistance of the 7079-T6 skins, new cracks typically nucleate in the skin at the corners of the riveted repair, leading to ever-larger repairs. Besides bonded crack patching of the damaged sections, the only remaining option is to selectively reskin the fleet. Because the rivet holes were hand-drilled during original manufacturing of the C-5A, some holes are too close to the edge to meet the minimum edge distance requirement after drilling out and reaming the existing holes. Reskinning will be very costly because most of the crown frames and stiffeners would have to be replaced as well. Because of the static ground loads on the crown, the reskinning operation requires complex “spider frames” to unload the crown section.

The price for reskinning has been estimated by San Antonio ALC personnel to be $5 million per aircraft. The cost per patch of bonded repairs in a “production environment”

has been estimated at $2,000. Thus, even if as many as 25 patches were needed per aircraft, the total cost of repairing the 76 "A" models would be less than the cost of reskinning one aircraft.

1.4. REQUIREMENTS FOR REPAIR METHODS

Repairing a fuselage skin is a good alternative when both economical and technical benefits can be achieved. In general, the highest cost of a repair will be downtime and labor. The best available repair method will decrease downtime of the aircraft in future operations and will also provide the safest way of operating aging aircraft. At the same time, the selected repair method should not cause new problems in the structure.

1.5. PROCESS (RIVETED/BONDED)

Most traditional repair methods use riveted patches over structural damage. The disadvantage of a riveted repair is the possibility of nucleating new cracks from the additional rivet holes. The high-strength 7079-T6 aluminum alloy used on the C-5A fuselage skin material is very susceptible to material defects and the residual stresses caused by riveting. A bonded repair provides smoother load introduction through the entire bond line instead of discrete stress concentrations at the rivets. A properly installed bonded patch is a durable repair that will outperform riveted repairs.

1.6. MATERIALS

The choice of the patch material is just as critical a variable as the bonding process in crack patching. Following the arguments from the last paragraph, one can see that a bonded repair can be preferred over a riveted repair from a mechanical standpoint.

This section discusses the importance of proper patch selection for a bonded repair. Aluminum and titanium were considered as candidates, as were the composites boron and graphite/epoxy and GLARE. Selected mechanical properties for the materials are summarized in table 1.

Table 1. Selected Properties of Patch Materials.

	2024-T3	Ti 6Al-4V	GLARE 2 (3/2)	Boron/epoxy	AS4 Carbon/ epoxy
E_{11} (GPa)	72.4	116	68	210	186
E_{22} (GPa)	72.4	116	50.7	25	12
α_{11}, 10^{-6}/°C	22.7	7.1	17.9	4.5	-0.9
$\sigma_{11\ yield}$ (MPa)	303	925	390	n/a	n/a
$\sigma_{11\ ultimate}$ (MPa)	485	1200	1100	1590	2913

Stiffness of the patch material

In general, the extensional stiffness (Et) of a patch will be equal to or slightly greater than the stiffness repaired parent material. Higher modulus patch materials are thus thinner than moderate- to low-modulus patches. For very thick sections or aerodynamically critical areas, the thickness of the patch is important. However, with relatively thin fuselage skins, the thickness of the patch is not important. This is especially true aft of the wing, where a thick boundary layer exists.

Care must be taken in design not to make a repair patch *too stiff*, to avoid attracting unwanted, additional skin loads into the repaired area. A high-strength, low elastic modulus material can work better in these cases, by restoring strength without substantially increasing local structural stiffness.

Coefficient of Thermal Expansion (CTE)

The highest loads in a pressurized fuselage typically occur at low temperatures. The difference in CTE between the patch material and the fuselage plays an important role in patching effectiveness. However, the effects of thermal expansion are complicated. This is due to the constraint placed on the (locally heated) repair area by the surrounding cooler structure during bonding. The constraint means the cracked skin cannot freely expand during the repair process.

For all patch materials, a stress free state between patch and fuselage skin is present at the cure temperature of the adhesive, typically 100 to 125 oC (200 to 250 oF). After cooling down to room temperature or to cruise altitude (-50 oC (-60 oF)), a moderate- or

high-CTE material will put the crack in compression. However, a low-CTE patch material will put additional tensile loads on the crack. References [2,3] describe this effect in detail. Because of this effect, a moderate- or high-CTE patch material is preferred over a low-CTE patch material when high skin stresses are encountered at low temperatures.

Strength of the material

Higher strength repair materials generally result in thinner patches. However, most high-strength materials tend to behave in a brittle manner. An ideal patch material would have high ultimate strength, yet exhibit limited yielding to warn of impending patch failure.

Surface Preparation

Surface preparation on an aircraft is substantially different from a factory surface pre-treatment. Phosphoric Acid Anodized (PAA) bonded surfaces have an excellent service history in the aircraft industry. Bonded aircraft structures using this surface pre-treatment technique have given over 30 years of durable service.

Some problems exist in using acid-containing surface pre-treatments in the field. Concern about entrapping acids in rivet holes and cracks has led to development of alternative field pre-treatments. A newer process using organic "silane" adhesion promoters has gained acceptance for in-field bonding in the past decade.

The silane surface preparation technique involves:

- grit blasting the aluminum skin,
- cleaning the skin with an organic solvent,
- applying a silane adhesion promoter, and
- priming and curing with an epoxy adhesive primer.

Accelerated laboratory tests were performed comparing the silane surface preparation to the PAA process. After optimization, the silane technique has shown the same excellent durability as the PAA process [6]. The silane pre-treatment has been further proven in previous installations on USAF aircraft (e.g., C-141 weepholes, F-16 fuel vent hole repair) over the last five years.

2. ANALYSIS

The bonded repair concept has been analyzed extensively to predict the performance of the repair system on aircraft. The program used to design the patches, CalcuRep, was developed over the past four years at Delft University and the U.S. Air Force Academy. CalcuRep analyzes the effects of different patch materials, cure temperature and flight conditions (stress levels, altitude, etc.) on the effectiveness of the repair. The closed-form analytical solution in the code is based on a mathematical model developed by Rose [1]. The model has been verified with a large body of experimental data and by comparing its results with detailed finite element analyses [2].

2.1. Repair Design and Analysis with CalcuRep

The CalcuRep code is much more efficient than Finite Element Models for designing and analyzing patches. A typical maintenance engineer can learn to use the program in a few hours and can design a suitable patch within half an hour on a PC. The results of the calculation for the two patches used to repair T/N 69-021 are given in section 4.

This section describes calculations regarding the influence of patch material and cruise temperature on K reduction at the crack tip (to slow down or stop crack growth) and adhesive shear strain (related to the durability of the bonded joint).

CalcuRep quickly calculates the important design parameters for a bonded repair:

- the repaired stress intensity factor at the crack tip,
- the maximum stress in the patch (over the crack),
- the maximum skin stress (at the patch tips),
- the maximum shear strain in the adhesive, and
- the shear load transfer length in the bond line.

The critical locations in a bonded repair are shown in figure 1.

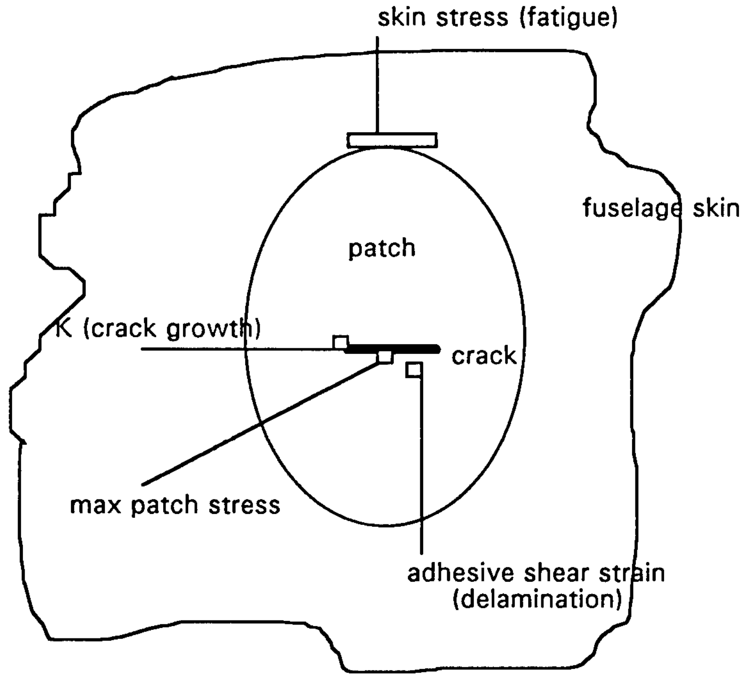


Figure 1. Critical locations in a bonded repair.

Each component is critical to a particular failure mode in the bonded repair system. For example, the *repaired stress intensity factor* at the crack tip determines how quickly a repaired crack will grow, influencing inspection intervals. The *maximum stress in the patch* over the crack helps select the patch material and thickness, while the *maximum skin stress* (at the patch tips) indicates whether a risk exists for new crack nucleation in the skin adjacent to a repair. A low *maximum shear strain in the adhesive* is important for good patch adherence and durability of the bond. Finally, the *shear load transfer length* ensures that the bond area is large enough for proper load transfer and damage tolerance.

Example CalcuRep calculations for 1.0 mm thick 2024-T3 aluminum test specimens discussed in the following section are shown in Table 2. The test conditions modeled were a fuselage skin at room temperature under 120 MPa of uniaxial tension.

Table 2. Summary of CalcuRep output.

Patch type / Design Criteria	boron/epoxy 3 plies, t = 0.39 mm	boron/epoxy 4 plies, t = 0.52 mm	GLARE 2, 3/2 lay-up, t = 1.1 mm
• K repaired, MPa$\sqrt{m}$	4.7	3.8	1.7
• max adhesive shear strain	0.044	0.035	0.016
• max patch stress, MPa	392	303	147
• max skin stress, MPa	195	200	201

At room temperature, the GLARE patch is slightly more effective than boron/epoxy in reducing the crack tip stress intensity factor K, while keeping adhesive shear strains substantially lower. Patch and skin stresses are acceptable for all configurations. At the much lower temperatures encountered at cruise altitude, the low CTE of boron will reduce its crack-slowing effectiveness (K will increase), while adhesive shear strains increase to more critical levels as well. The more closely matched CTE of GLARE means that it remains more effective than boron at low temperatures.

3. EXPERIMENTAL RESULTS

3.1. FATIGUE TESTING OF SMALL UNSTIFFENED PANELS

To verify the predictions of the CalcuRep code, several fatigue specimens matching the modeled configuration were built and tested. Coupon tests were performed pre-cracked 2024-T3 aluminum panels repaired with boron/epoxy and GLARE 2. The configuration of the test panels was:

- Panel width 152 mm, length 400 mm
- Fatigue pre-crack length 25 mm
- Uniaxial loading, 6-120 MPa (GLARE)
12-120 MPa (boron/epoxy)
- Adhesive AF163-2, 0.13 mm thick
- Cure temperature 120 °C (250 °F)
- Patch size (optimized by material)
60 mm long x 50 mm wide (GLARE)
75 mm long x 50 mm wide (boron/epoxy)

The boron/epoxy was tested at a slightly lower stress amplitude (6 MPa higher minimum stress than GLARE). This was due to the large residual curvature induced in the test panel due to the large CTE mismatch between boron and aluminum. This five percent lower stress amplitude should slightly favor the boron patch performance over the GLARE in terms of slower crack growth.

Figure 2 shows the results for an unpatched panel and both GLARE and boron patched panels. Both patches extend the fatigue life of the panel considerably over the unpatched case. However, the GLARE patched panel gave by far the longest fatigue life, showing no crack growth in the first 400,000 cycles, followed by the slowest crack growth rate of any of the repairs. The better crack-stopping performance of the GLARE versus the boron patch was predicted by the CalcuRep analysis (lowest K, lowest adhesive shear strains) and can be attributed to the better CTE match between GLARE and aluminum. By contrast, the boron repair suffers from a higher repaired K and higher adhesive shear strain values, allowing immediate crack growth, even with a lower applied stress amplitude.

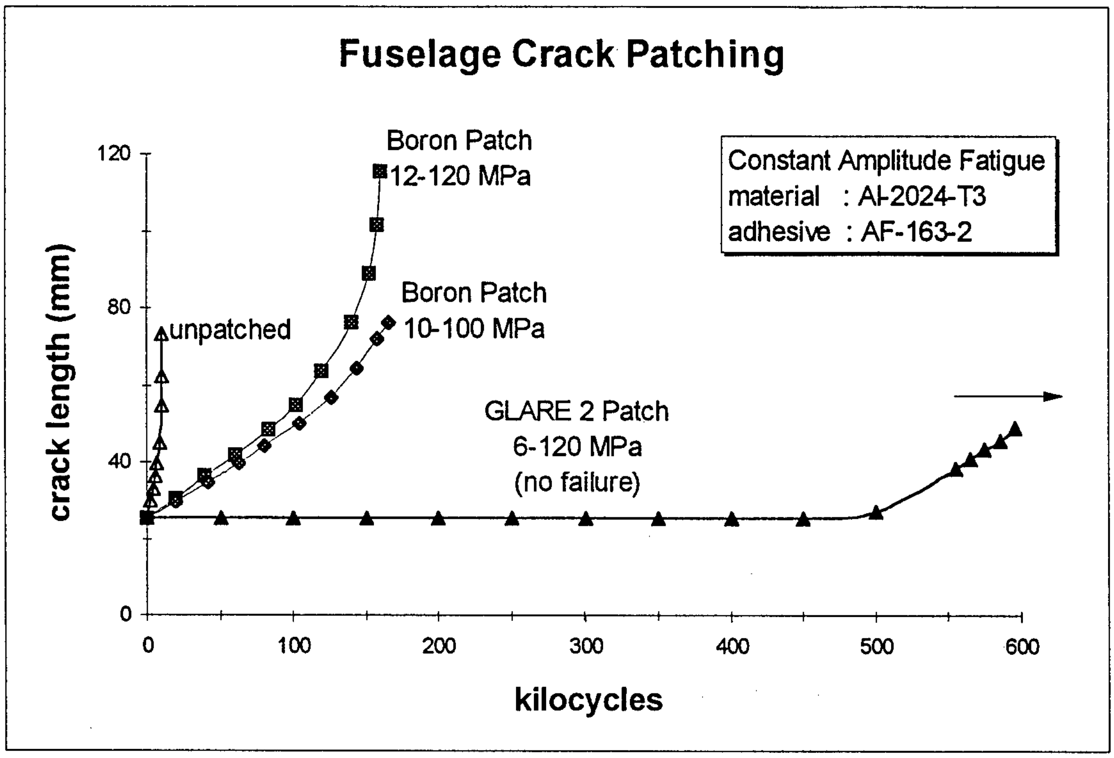


Figure 2. Constant amplitude fatigue test results.

4. DESIGN OF THE C-5A CROWN REPAIR

4.1. CALCUREP CALCULATIONS FOR ALL LOAD CASES

The two cracks repaired in T/N 69-021 are located at fuselage station 1675 near stringer 78 and at fuselage station 1784 near stringer 70. To design a suitable patch, the configuration has to meet the guidelines for typical loads, limit load and ultimate load. At typical loads the repair must show durable behavior, and greatly slow crack growth. At limit load, no yielding of any component is allowed. At ultimate design load (equal to 1.5 times the design limit load), the structure shall not fail. which means that all stresses should be less than the ultimate strength of the respective components.

Table 3 gives the design limits of the critical design variables for three different load cases (typical, limit, and ultimate).

Table 3. Design limits for bonded patches

	Typical loads	Limit load	Ultimate load
K, MPa√m	da/dN < 10^{-4} mm/cycle	< Kc/2.25	< Kc/1.5
Patch stress, MPa	< 0.5 σ_{yield}	< σ_{yield}	< $\sigma_{ultimate}$
Skin stress, MPa	< 0.5 σ_{yield}	< σ_{yield}	< $\sigma_{ultimate}$
adhesive shear strain	< 0.5 γ_{yield}	< γ_{yield}	< 1.8 γ_{yield}
Patch length/adhesive load transfer length	40	40	40

For the typical repair configuration on a 7079-T6 crown panel, this results in the guidelines given in table 4.

Table 4. Design limits for bonded GLARE 2 patches on 7079-T6 skins

	Typical loads	Limit load	Ultimate load
K, MPa√m	<4 MPa√m	<22 MPa√m	< 33 MPa√m
Patch stress, MPa	<180	<360	<1100
Skin stress, MPa	< 213	<426	< ultimate
adhesive shear strain	< 0.0446	<0.1070	< 0.1606
Patch length/adhesive load transfer length	40	40	40

The design of the GLARE patch has to meet as many as possible of the given design given in table 5. The CalcuRep input file used to design the patches consisted off the following data:

Table 5. CalcuRep input file for patches @ FS 1675 & FS 1784

	FS 1675	**FS 1784**
Skin thickness	1.3 mm	1.3 mm
Crack length	12 mm	22 mm
Frame spacing	628 mm	508 mm
Stringer spacing	155 mm	155 mm
Long stress typical*	121	121
Hoop stress typical	100	100
Long stress @ limit	223 MPa	227 MPa
Hoop stress @ limit	81 MPa	83 MPa
Patch	GLARE 2, 3/2 lay-up (1.4 mm thick)	GLARE 2 4/3 lay-up (1.55 mm thick)

* Load occurs 500 times per 1000 flight hours (15000 per life)

Heat blanket 275 by 275 mm
Cure temperature 120 C
Cruising altitude: 10,000 m typical, 1000 m limit/ultimate
Taper thickness 0.3 mm
Patch length: 100 mm
Patch width: 90 mm
Ultimate load: 1.5 * Limit

The results of the calculations for FS 1675, stringer 78 are given in table 6.

Table 6. Calculation results for FS 1675

Parameter	**Typical (121/100)**	**Limit (223/81)**	**Ultimate (335/122)**
K, MPa$\sqrt{m}$	3.03	7.13	19.6
Patch stress, MPa	148	268	394
Skin stress, MPa	195	352	517
adhesive shear strain	0.0343	0.0805	0.1523
Patch length	153	153	153

Comparing the calculated results with the table for the design limits shows that all design criteria have a positive margin of safety except for the suggested patch length (100 mm instead of 153 mm). The patch length is 26 times the load transfer length, which should still provide a good enough resistance to creep of the adhesive, and was preferable to bridging the butt joint near the crack.

Table 7. Calculation results for FS 1784

Parameter	**Typical (121/100)**	**Limit (227/83)**	**Ultimate (341/125)**
K, MPa√m	2.67	6.98	20.7
Patch stress, MPa	133	246	363
Skin stress, MPa	199	364	533
adhesive shear strain	0.0276	0.0723	0.1361
Patch length	157	157	157

Once again, all design criteria have a positive margin of safety except for the suggested patch length (101 mm instead of 157 mm), again due to the proximity of the repair to a butt joint in the skin. The patch length is again about 26 times the calculated load transfer length, which should be sufficient to prevent creep and ensure good patch durability.

4.2. FITTING THE PATCH TO STRUCTURE

Once the patch has been analyzed and roughly optimized using CalcuRep, further consideration must be given to the existing structure. Good design practice suggests to avoid rivets at the patch runouts (tips), because in those areas, the skin and the adhesive are higher loaded. CalcuRep allows the proper patch design, but the final outline is also largely affected by splice, doubler, stringer and frame configuration and the existing rivet pattern.

Figure 3 shows a trace of the existing rivet pattern and the patch location on the aircraft at FS 1700. At the critical areas (patch tips) the rivets are all covered by the patch. By covering the rivet a high stress concentration due to the patch tip and the rivet is prevented, improving the durability of the repair system. Figure 4 shows the location of the two bonded repairs.

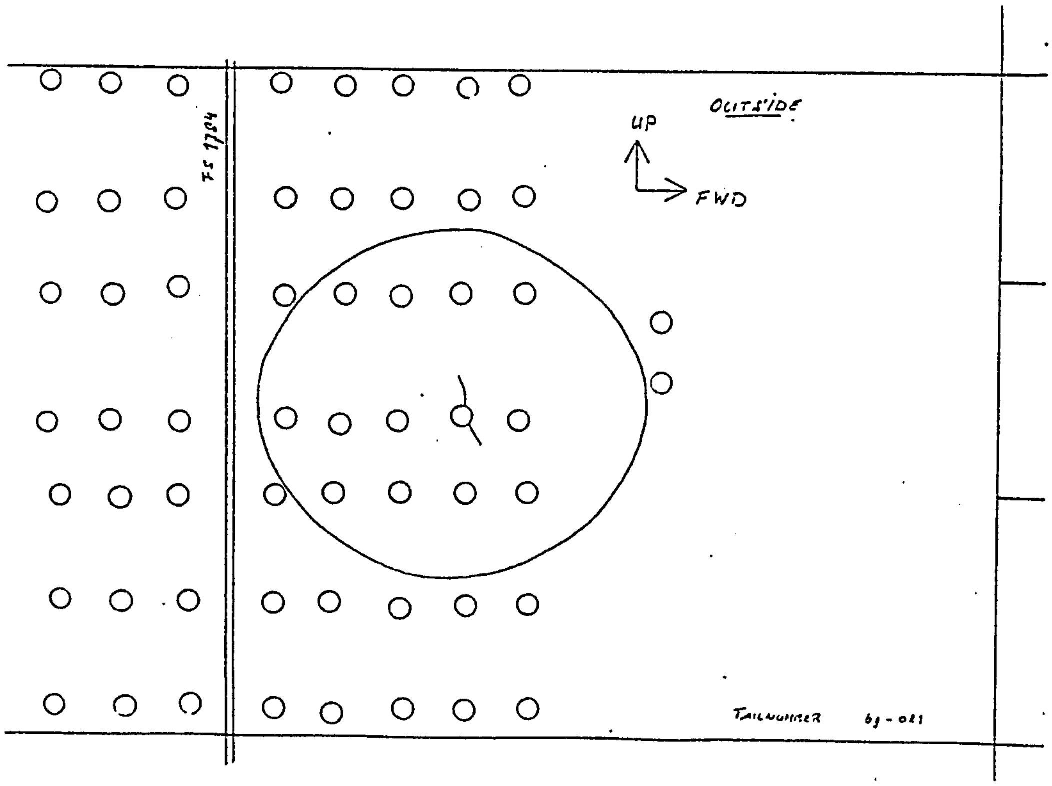
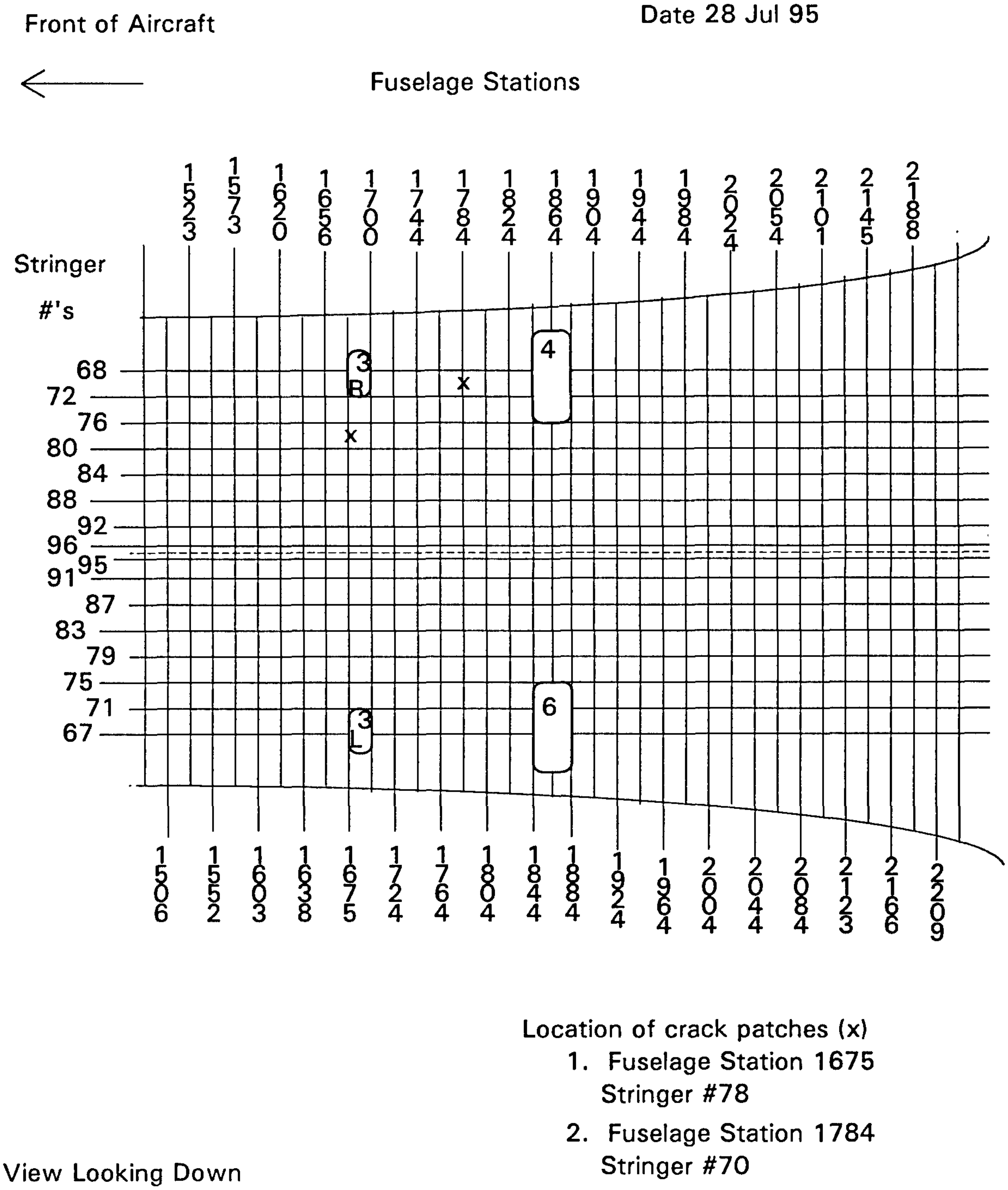


Figure 3. Trace of rivet pattern and patch location for repair to FS 1700.

Upper Aft Skin Cracks Worksheet

Aircraft Tail # 690020
Inspector Larivee
Unit/Phone 589-2024
Date 28 Jul 95



Location of crack patches (x)

1. Fuselage Station 1675
   Stringer #78
2. Fuselage Station 1784
   Stringer #70

1. Note location of all cracks and patches found.
2. Return completed worksheet to SA-ALC/LADS, Kelly AFB, TX.

Figure 4. Location of fuselage cracks on T/N 69-021.

5. INSTALLATION

5.1. THERMAL SURVEY

Before the surface pretreatment can begin, a thermal survey must be conducted to locate any potential thermal problems (e.g., heat sinks) created by the substructure under the fuselage skin. Since frames and stringers will absorb more heat, good thermal control is need to assure proper curing in all areas. Extra heat blankets or insulating the substructure can decrease the problem.

The thermal survey of the repair locations showed no big problems in heating the structure. A 20 °C fluctuation was considered acceptable. Cure times were then adjusted to match the lowest cure temperature. A computer controller monitored and adjusted the heating to achieve proper temperatures. An even temperature distribution allows a higher average cure temperature and therefore a faster cure cycle in the complete section. The thermal survey took approximately three hours.

5.2. SURFACE PREPARATION

The surface preparation used can be described with the following steps.

1. Degrease the aluminum surface with MEK
2. Abrade the aluminum surface using a Scotch Brite pad wetted with MEK
3. Clean the surface again with MEK
4. Grit blast with Al_2O_3 (50 micron) using dry nitrogen as pressure medium
5. Blow off the grit using dry nitrogen
6. After the silane solution has been hydrolyzed (stirred in water) for at least one hour, apply the silane solution to the aluminum skin. Keep wetted for at least ten minutes.
7. Cure the silane @ 93 °C (200 °F) for one hour
8. Apply BR127 adhesive primer
9. Cure @ 120 °C (250 °F) for one hour
10. Proceed with bonding

The surface preparation lasts approximately eight to ten hours. The only allowable break in the process may be between steps 9 and 10. Stopping at any other time means the process must be completely re-accomplished. The critical parameters in the bonding process are the water used to hydrolyze the silane (pH = 5), the silane

concentration (1 to 2%), the stirring time of the silane (1 to 15 hours) and the silane cure temperature (93 °C [200 °F], 1 hour).

5.3. PATCH MANUFACTURING

The most efficient shape for a reinforcement is an ellipse, which is traced onto a piece of paper and transferred to the GLARE sheet material. A small sheet of GLARE is then cold-rolled to match the diameter of the fuselage. A shear cutter can be used to cut it to the rough dimensions, and a belt sander can be used to trim the patch to the final outer dimensions.

To decrease secondary bending effects in the skin, the patch must be tapered. The minimum taper thickness is 0.3 mm. Too thin a minimum taper thickness will reduce the ability to make a good bond fillet and risks damaging the edge of the patch. The taper angle should range be about 1:10. The taper angle is defined by drawing a smaller ellipse on the patch and filing the material from that line until the last aluminum layer of the patch.

Making a taper in a GLARE patch is easy to perform because the different pre-preg and aluminum layers are easily recognizable. The pre-preg layers will show up as green layers, while the aluminum layers have the natural silver color. Handling the patch while making the taper is easier if the patch is stuck to a wooden template with double sided tape.

After manufacturing to the proper shape, cleaning/degreasing the patch with a solvent like MEK or isopropyl alcohol makes the patch ready to be bonded to the fuselage skin.

The patch can be made anytime during or before the process. Making the patch takes about two hours by hand. In a "production" environment, manufacturing could be speeded using numerically controlled milling.

5.4. BONDING

After the surface preparation and patch manufacturing are complete, the area to be bonded should be kept free of contaminations by covering and sealing it with a plastic film. (If the surface is contaminated after curing the adhesive primer, it can be cleaned by simply wiping it with MEK.) The patch can now be bonded in place.

1. Remove the adhesive from the freezer and allow it to thaw to room before removing it from the plastic bag.
2. Clean the shaped patch with MEK. No further surface pre treatment is needed, because the patch is already anodized and primed with BR127.
3. Apply the adhesive film it to the back side of the patch. Take care to prevent entrapping air and rub all air bubbles to the edge of the patch with a roller.
4. Build the vacuum bag.
5. Apply full vacuum (22-24" Hg) for 30 minutes to debulk the adhesive before applying heat.
6. Decrease the vacuum to 16" Hg.
7. Heat up at a rate of 2-4 °C/minute (4 to 7 °F/minute) to 125 °C (250 °F).
8. Cure @ 125 °C (250 °F) for one hour.
9. Cool down at a rate of 4-6 °C/minute (7-11 °F/minute) to 50 °C (120 °F) before releasing vacuum.
10. Remove the bonding and vacuum equipment.

The bonding process takes approximately two to three hours.

6. INSPECTION METHODS

Once the patch has been applied, both the quality of the bond line and the crack must be monitored on a regular basis.

6.1. INSPECTION FOR CRACKS

The extremely slow crack growth expected under the patch can be monitored by using off-the-shelf eddy current techniques. Eddy current is a standard technique used to find cracks around fastener holes and is widely used in the aircraft industry. By using lower frequencies the crack can be measured through the thickness of the GLARE patch.

The method is based on creating an electromagnetic field in a probe. This field creates an opposite magnetic field in the conductive aluminum. This opposite field effects the field in the probe and indirectly its impedance. With a crack present in the Aluminum the opposite field will change and the Eddy scope makes this visible on the display.

6.2. BOND LINE INSPECTION

The most accurate method of ensuring the quality of the bond line against voids or disbonds is a sonic or ultrasonic transmission-based pulse-echo method. The most commonly used types are ultrasonic. The method is based on transmitting a wave through the bonded structure (both the patch and the skin). When there is a perfect bond, the biggest sound reflection will come from the back side of the skin. When there is a disbond or void in the bond line, the back side reflection comes earlier from the patch, and hardly any signal can transmit through to the skin. At least two off-the-shelf pieces of test equipment are available: the Fokker bond tester and the Stavely Bondmaster. The practical use of this equipment must be investigated. Wright Laboratory has several programs ongoing in this area.

A simpler but rather effective inspection method is a visual inspection of the adhesive fillet and the bleeder. When the fillet looks smooth and the bleeder has a moderate

amount of adhesive in it all the way along the bond line, the bond should be acceptable. This was done with both installations on T/N 69-021, and the results were good.

A coin tap check can also give a qualitative idea of the bond line quality. A coin tap is based on the same principles as the previous described ultrasonic inspections, and was successfully performed on both patch installations on T/N 69-021.

7. RECOMMENDATIONS

7.1. PROCESS IMPROVEMENTS

The longest times in the patch installation process are the three cure cycles involved (silane, primer, and adhesive). To decrease the total cure time, a program should be started at WL/MLSE to investigate the co-cure of silane and primer, primer and adhesive, or all three components. Experiments including wedge edge tests under accelerated aging ("hot/wet") are relatively simple and cheap to perform. Once an accelerated process is apparently acceptable, it can be tested using a limited number of fatigue specimens.

7.2. NON-DESTRUCTIVE INSPECTION TECHNIQUES

A standard for on-aircraft eddy current monitoring of crack growth is being developed at the Air Force Academy and will be available before the aircraft completes PDM. Development of applicable ultrasonic bond line techniques to monitor bond line quality should be done at WL/ML.

7.3. FLEET-WIDE IMPLEMENTATION

The authors recommend strongly that this repair be considered for fleet-wide implementation. The price for reskinning the crowns of the 76-aircraft fleet of A models has been estimated by SA-ALC engineers at $380 million. The cost of bonded repairs to the fleet should not exceed $4 million.

For fleet-wide repairs, a two-team approach is suggested. If a specialized depot team were trained to perform the repairs on all aircraft scheduled for PDM in the next two to three years, a qualified contractor team could repair the remaining aircraft at their main operating bases over a one to two-year period.

8. SUMMARY AND CONCLUSIONS

Two bonded GLARE composite patches were successfully installed over two fuselage skin cracks in C-5A T/N 69-021 by a team of San Antonio ALC, Wright Laboratory, and USAF Academy engineers. The repairs represent a breakthrough cost-effective solution to the problem of widespread cracking in the aft fuselage crown and can extend the use of C-5As at about 1% of the cost of reskinning the fleet. The bonded repairs prevent cracks from growing to a critical length, where retirement of the aircraft or expensive panel replacements would be the only other options.

This repair should be considered for fleet-wide implementation using a two-team approach. The approach would involve a specialized depot team for repairs on all aircraft scheduled for PDM in the next three years and a qualified contractor team to repair the remaining aircraft at their main operating bases over a one to two-year period.

9. REFERENCES

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